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Technical Report No. 32-335

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CONTENTS

TABLES

FIGURES

ABSTRACT

This Report describes the conception, design, development, and operation of the midcourse trajectory-correction propulsion system utilized in the *Ranger* spacecraft to facilitate lunar impact. The propulsion unit consists of a small monopropellant-hydrazine-fueled rocket of 50-lb vacuum thrust capable of delivering a variable total impulse in conjunction with an integrating accelerometer system. Functionally the rocket is of the pressure-fed constant-thrust type. Injection pressure is obtained from compressed gas, either helium or nitrogen, which passes through a pressure regulator and forces the propellant from a bladdered tank to the rocket engine. The rocket engine contains a quantity of catalyst to accelerate the decomposition of anhydrous hydrazine. Engine ignition is accomplished through the injection of a small quantity of a hypergolic oxidizer, nitrogen tetroxide. All valving functions for the propulsion unit are accomplished with explosively actuated valves.

In-flight performance of the unit as a portion of the *Ranger 3* and *Ranger 4* missions is described.

I. INTRODUCTION

The exacting objectives of this nation's lunar and planetary exploration program have presented the propulsion discipline with new and unusual areas for propulsion system applications. Within these applications areas are devices to accomplish post-injection guidance maneuvers. Such systems may be more specifically classified as being used for midcourse maneuvers or approach correction maneuvers, depending upon their point of application. A midcourse maneuver may be viewed as a single impulse or perhaps a series of small impulses made relatively early in flight to eliminate or reduce errors introduced by the injection guidance system and/or by the performance of the boost propulsion system. However, even if the desired trajectory is thereby obtained, additional corrective maneuvers may be required to adjust this flight path because of the current uncertainties in astronomical measurements and because of the additional possibility of introducing new guidance and propulsion errors while the midcourse maneuver is being made. Therefore, another correction maneuver known as an approach correction is envisioned for many missions. These maneuvers would be expected to be made relatively late in flight, and in addition to correcting the trajectory would serve to place the spacecraft in a proper position to undertake a terminal maneuver.

The results of previous investigations (Ref. 1) have led to some interesting conclusions regarding the characterization of propulsion systems for such applications. These were: (1) that the size and thrust level of devices for such applications are relatively low, with thrust-to-weight ratio approximately 0.1 for current spacecraft, and decreasing to a lower ratio in future spacecraft with more advanced accelerometers and integrators; (2) that, from the standpoint of size, the propulsion system would usually constitute 5% or less of the gross spacecraft weight; and (3) that some form of post-injection guidance maneuvers is required for virtually all spacecraft currently envisioned. Hence it appears evident that this application area presents a challenge to the propulsion system designer to advance the state-of-the-art in the specification of propellants, in the design of components, and in the integration of the propulsion system into the spacecraft from both a ground operations and a flight standpoint.

This Report describes one such propulsion system which was developed as a portion of the *Ranger* program. The basic purpose of the *Ranger* program is the advancement of space science and technology through the initial investigation of the surface of the Moon. At the time of initiation of this Report the over-all program consisted of three phases. The first phase comprised *Rangers 1* and *2*, which had the objective of proving the design integrity and flight mission capability of the spacecraft. The second phase involved *Rangers 3, 4,* and *5*, whose missions were to rough-land an instrumented capsule on the surface of the Moon. The third phase of the program involves *Rangers 6* through *9* and is also a series of lunar impacts; this phase will be devoted to a high-resolution television investigation of the lunar surface and terrain.

The propulsion system to be described in this Report is included in all *Ranger* spacecraft with the exception of the first two. In order to understand the propulsion system and the restraints under which it was designed and must operate, it is desirable to describe the overall *Ranger* mission in some detail.

The *Ranger* system utilizes an *Atlas* booster for its first stage, an *Agena B* second stage, and the JPL *Ranger* spacecraft. The basic configuration of the *Ranger 3, 4,* and *5* spacecraft consists of a hexagonal frame containing the electronic packages; two erectable solar-power panels; a separable lunar landing capsule containing equipment for the lunar experiments; and a movable parabolic high-gain communications antenna (Fig. 1).

The over-all flight sequence of events is shown in Fig. 2. The spacecraft is initially confined within a shroud for environmental protection during the launch phase. The shroud is ejected following the *Atlas* sustainer burnout. At the conclusion of the first *Agena* burn, the spacecraft is in a coasting or parking orbit. A second ignition and burn of the *Agena*, concluding in spacecraft injection, is followed by the separation of the spacecraft from the *Agena B*.

After separation, the spacecraft's Sun and Earth acquisition sequence is initiated. The attitude-control system is activated, the solar panels are erected, and the high-gain antenna is rotated to a preset hinge angle. Solar sensors controlling the attitude-control jets cause the spacecraft to point its roll axis toward the Sun, thus placing the solar cell power system in operation. The spacecraft then turns about the roll axis until the high-gain antenna beam lies in the plane defined by the spacecraft roll axis and the Earth. Maintaining the antenna beam in this plane, the Earth sensors command the antenna to move so that its propagation axis intersects the Earth, establishing the high-gain communication link. The spacecraft then proceeds to coast in the attitude of Sun and Earth acquisition.

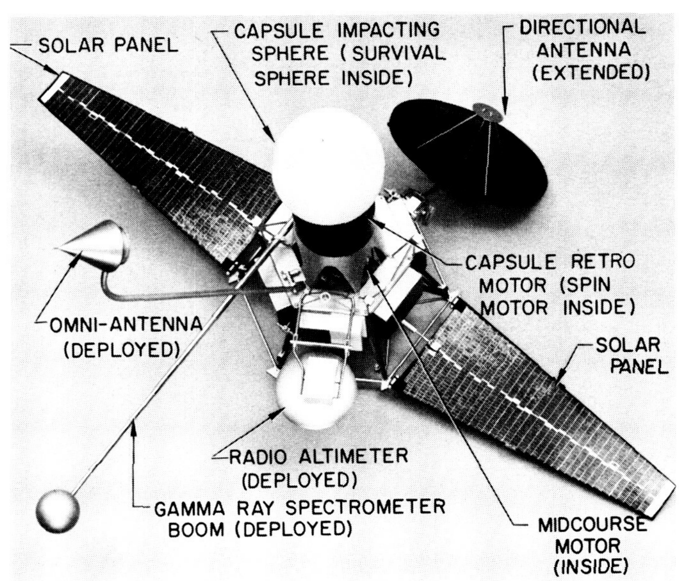

Fig. 1. *Ranger* spacecraft shown in terminal maneuver mode, just prior to capsule separation

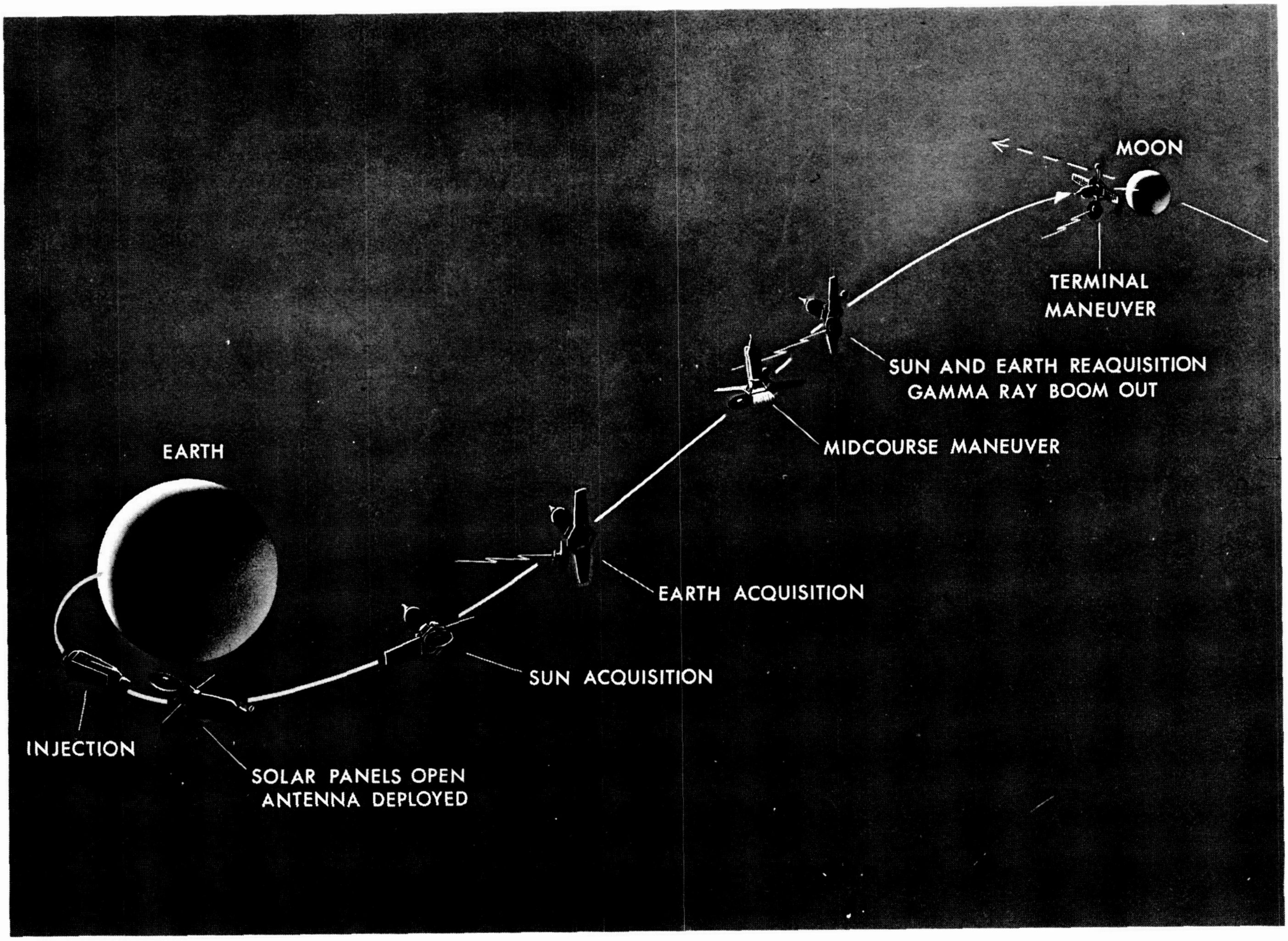

Fig. 2. ***Ranger 3*** **flight plan**

After a suitable tracking period, approximately 16 hr after launch, the required trajectory corrections are computed and the corrective maneuver commands are transmitted to the spacecraft. The resulting midcourse maneuver (see Fig. 3) turns the spacecraft to a prescribed angle, supplies the necessary thrust correction by firing the midcourse propulsion unit, and then returns the spacecraft to its Sun and Earth orientation. Upon approach to the lunar surface, the terminal maneuver is performed to align the vidicon camera for high-resolution pictures of the Moon, and to orient the lunar landing capsule for its subsequent separation and retro-braking. Commands from the Earth initiate the terminal maneuver.

Upon a signal from the radio altimeter, the capsule spin-motor fires, simultaneously spinning the capsule and lifting it out of its support structure approximately 2 ft. At this time, the capsule retro-motor fires and reduces the capsule approach velocity of approximately 9,000 ft/sec to essentially zero in 10 to 12 sec. The spacecraft plunges along a trajectory separate from that of the capsule, impacting the Moon.

The design of the midcourse propulsion system actually represents the culmination of thoughts and designs generated since the publication in 1958 of Ref. 2, in which liquid monopropellant propulsion devices were analyzed and proposed for the velocity control of early satellites and space probes. Specific results of this continuing effort were the development of a small propulsion device for vernier velocity adjustments, shown in Fig. 4 and described in Ref. 3, 4, and 5, and design studies of velocity control steering motors for the *Juno IV* vehicle, Ref. 6.

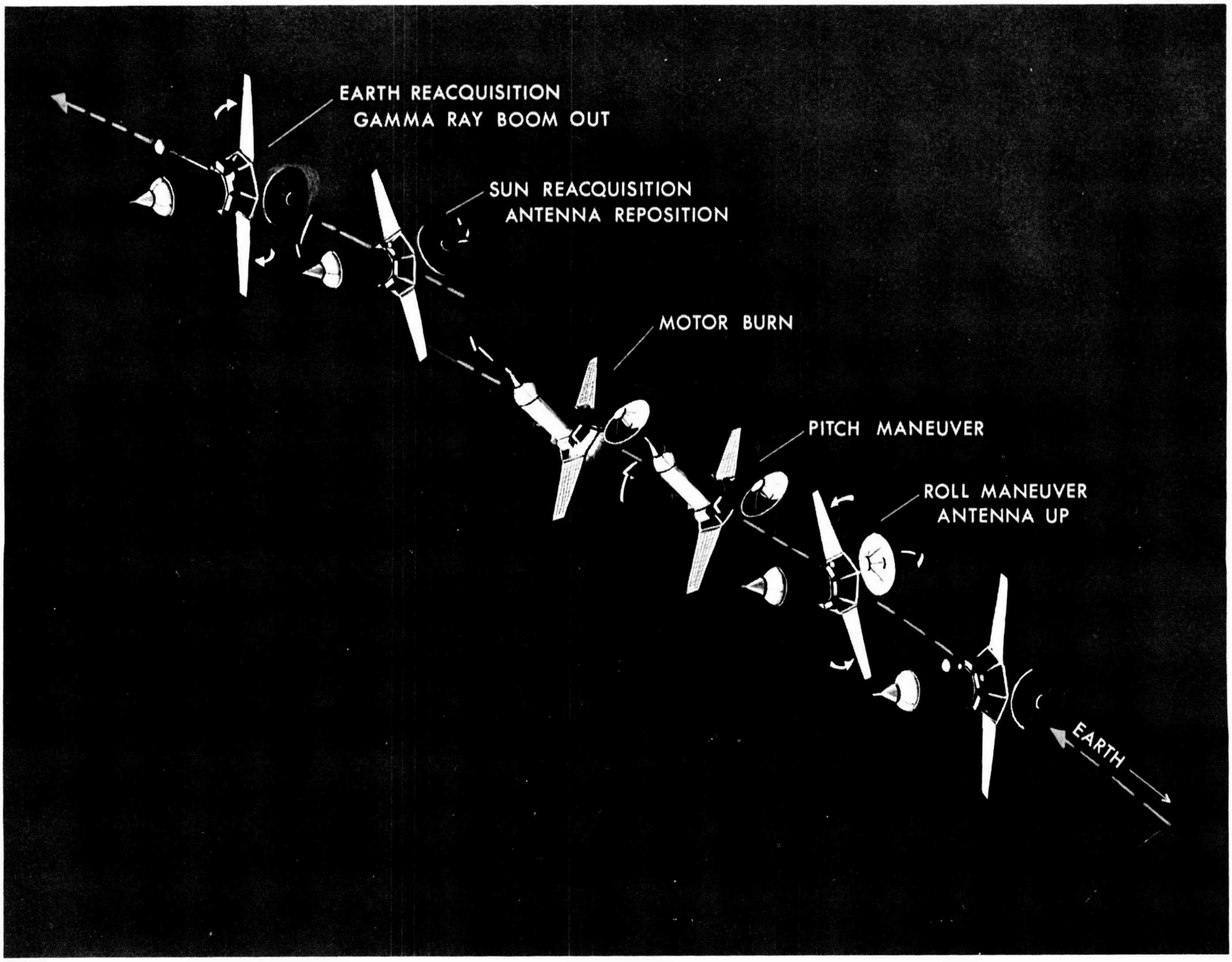

Fig. 3. *Ranger 3* midcourse maneuver

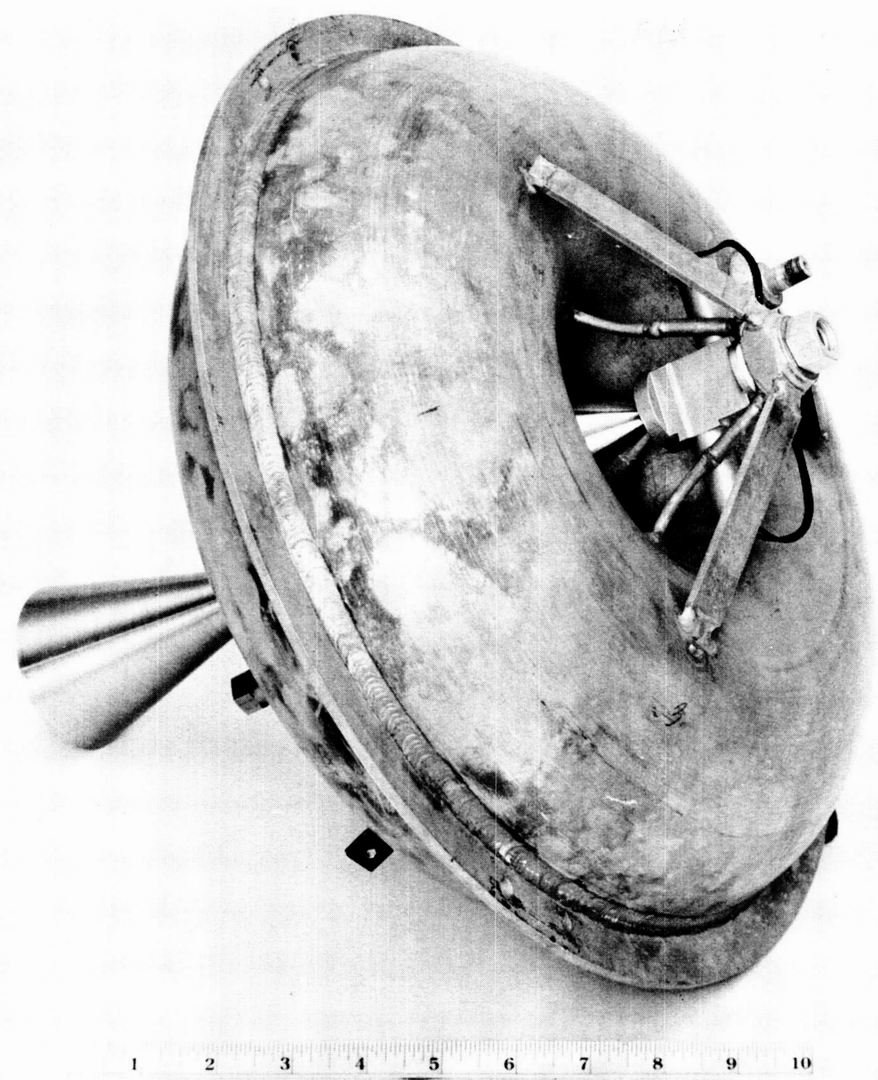

Fig. 4. Monopropellant-hydrazine velocity vernier rocket, 15-lb thrust

II. SYSTEM DESCRIPTION

The *Ranger* midcourse propulsion system, shown in Fig. 5 and described schematically in Fig. 6, consists of a small monopropellant-hydrazine-fueled propulsion system of 50-lb vacuum thrust. The decision to employ this type of propulsion system was based upon the simplicity of the system, its high degree of flexibility with regard to total impulse, its adaptability to vehicle design, and its general state of development. It is functionally a pressure-fed constant-thrust rocket. Injection pressure is derived either from compressed helium or nitrogen gas, which passes through a pressure regulator and forces the fuel from a bladdered propellant tank into the rocket engine. The rocket engine contains a quantity of particulate catalyst to accelerate the decomposition of anhydrous hydrazine. Engine ignition is accomplished through the injection of a small quantity of a hypergolic oxidizer, nitrogen tetroxide. Three simultaneously operating, explosively actuated valves are required to initiate operation of the system and two simultaneously operating explosive valves are required to terminate system operation.

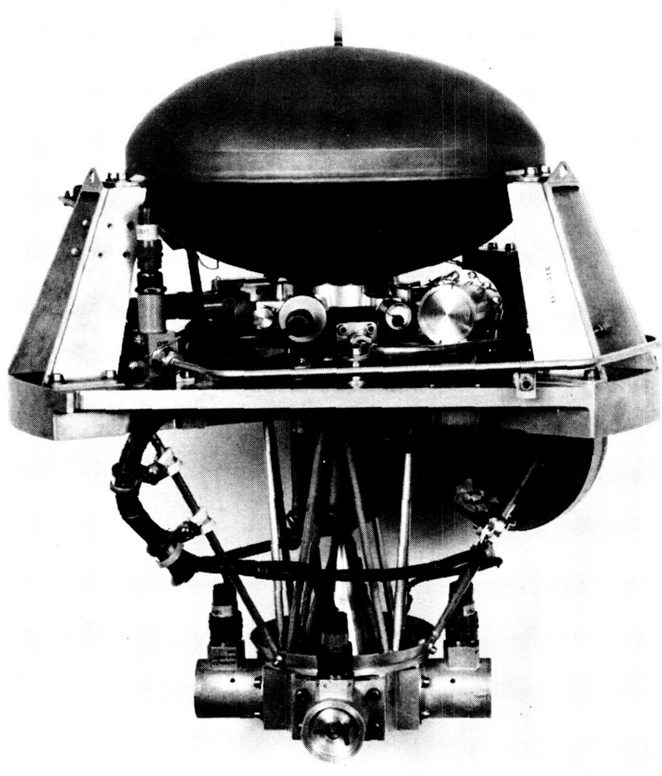

Fig. 5. *Ranger* midcourse propulsion unit

The unit is capable of delivering a variable total impulse to the spacecraft via command of an integrating accelerometer system. The duration of engine operation is determined by comparison of a ground-commanded velocity increment, determined from tracking data, with a velocity increment computed by an on-board integrating accelerometer system.

In general, the design and operational philosophy of the system were predicated upon the utilization of the highest performance monopropellant system available consistent with present technology. Considerable effort was expended to maximize system reliability and reproducibility, to minimize pre-flight handling and interaction with other sub-systems on the spacecraft, to minimize in-flight electrical signals to the propulsion system, to avoid electrical or mechanical sequencing, and to minimize the number of components. Underlying the program was the general aim of utilizing as much as possible the techniques, methods, and components previously developed for storable liquid propellant systems programs.

The following design requirements for the midcourse propulsion system were a result of spacecraft physical and operational restraints, *Agena* injection guidance accuracies, and ground and in-flight environment conditions.

1. The unit must have the capability (by virtue of the propellant tank size) of imparting to a 750-lb-mass spacecraft a velocity increment of 120 ft/sec.
2. The unit must be capable of vacuum environment storage in excess of 50 hr.
3. The unit must ignite and operate in a hard vacuum environment.
4. The unit must ignite and operate in a gravitationless environment.
5. The propulsion system must be capable of one ignition and one termination.
6. The rocket-engine nominal thrust shall be 50 lb (vacuum) and shall be predictable to within $\pm 5\%$.
7. The propulsion system shall be capable of operating between +35 and +165°F.
8. The spacecraft shall be essentially stationary in attitude (not spinning).
9. The volume within the spacecraft available to the propulsion system shall consist of the volume bounded

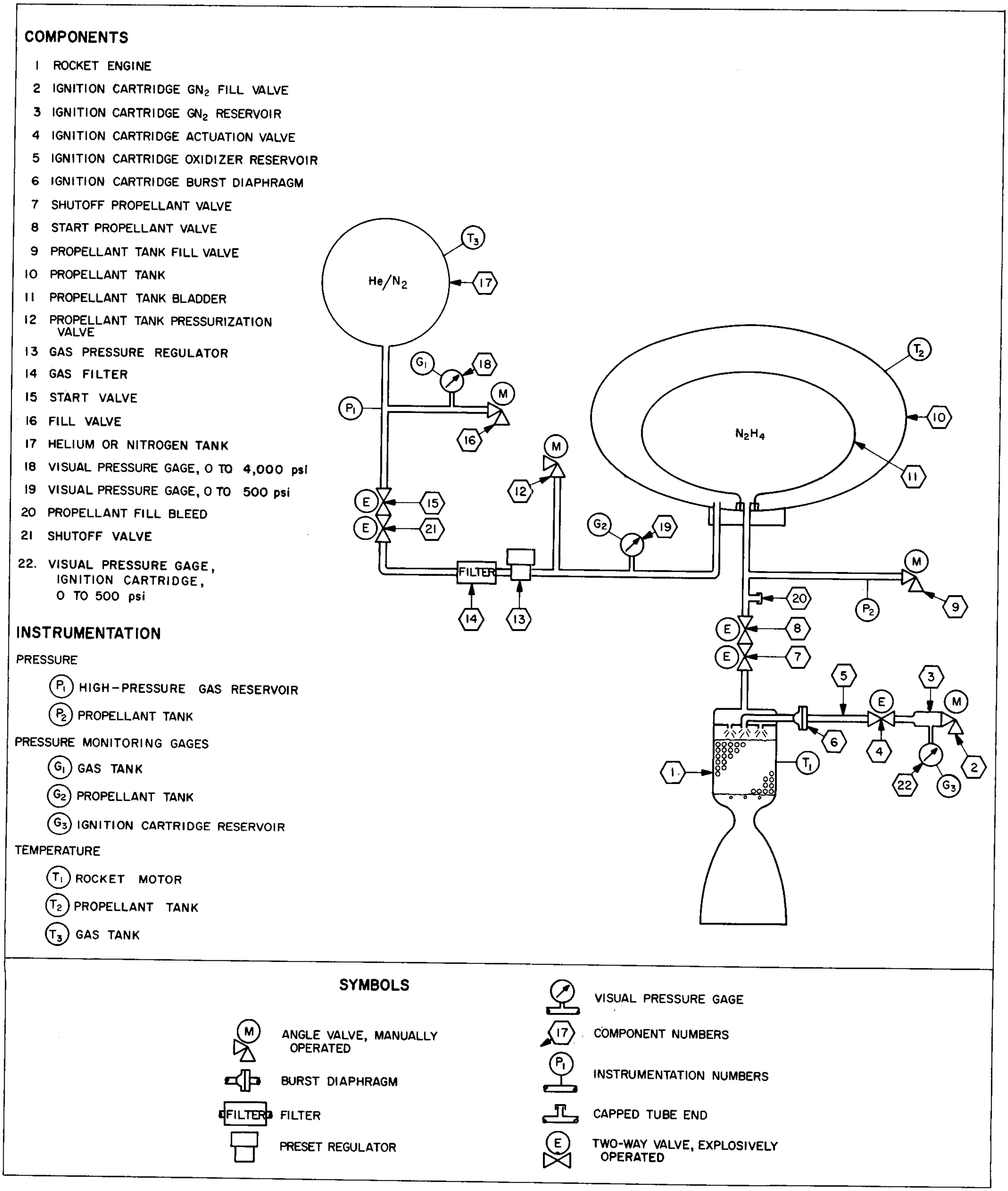

Fig. 6. Schematic drawing of *Ranger* midcourse propulsion system

on the sides by the internal hexagonal structure supporting the electronic chassis; on the top by the upper surface of the hexagonal compartment (capsule mounting plane); and on the bottom by an imaginary plane several inches below the lower surface of the hexagonal compartment consistent with the closed position of the high-gain antenna (see Fig. 1).

10. The thrust axis of the rocket motor must nominally coincide with the spacecraft roll reference line and be laterally adjustable upon assembly within a 2-in.-diameter circle about this reference line.
11. The effective thrust vector shall be predictable to within 0.1 deg angular displacement and ¹⁄₁₆-in. lateral offset of the geometrical motor centerline.
12. Thrust vector control shall be provided by four jet vanes capable of a maximum pitch and yaw restoring moment about the vehicle center of gravity of 3.2 ft-lb and a minimum roll moment of 0.1 ft-lb.
13. The entire propulsion unit in a dry condition shall be capable of heat sterilization consisting of 24 hr at 257°F.

A nominal engine performance summary of the *Ranger* propulsion system is given in Table 1, and nominal system pressures and temperatures encountered in its operation are shown in Tables 2 and 3, respectively.

Table 1. Nominal engine performance summary

Item	Value
Vacuum specific impulse I_{vac}, lb_f sec/lb_m	
Without jet vanes[a]	235.05
With 4 jet vanes deflected 10 deg[b]	231.76
Vacuum thrust F_{vac}, lb_f	
Without jet vanes	50.71
With 4 jet vanes deflected 10 deg[b]	50.00
Vacuum thrust coefficient $C_{F_{vac}}$[c]	
Without jet vanes	1.7558
Characteristic velocity c*, ft/sec[c]	4,306
Flow rate, lb/sec	0.2157
Nozzle throat area, in.²	0.15
Stagnation chamber pressure, psig[c]	189.1
Expansion ratio ϵ	44:1
Thrust vector capability, deg	
With vanes deflected 25 deg	±4.5

[a]Jet vane drag in the null position is negligible; therefore, engine data with null position jet vanes and engine data without jet vanes are essentially equal.
[b]Jet vane maximum deflection capability is 25 deg.
[c]Based on actual steady-state hot throat area (1.82% larger than ambient cold throat area).

Table 4 contains a weight breakdown of the propulsion unit. The tank weights reflect the fact that the design contains a 2.2 safety factor (i.e., burst pressure is 2.2 times the maximum working pressure at the maximum temperature, 165°F) so that personnel may work around the unit while it is completely pressurized. The loading density of the package is low because of the small quantity of fuel involved. For larger propellant quantities, many of the system parts would remain at the same weight, thus improving the loading density considerably.

Because of the simplicity of the system design, the unit can be prepared for flight in a minimum of time and without elaborate checkout procedures. The specific detailed procedures for assembly, test, and flight preparation are contained in formal JPL specifications and procedures; such detail is not felt to be appropriate to this Report. It is interesting, however, to review generally the events which comprise the pre-flight preparations of the system. The general sequence of such events may be considered to start with a fully fabricated and checked-out system which has successfully passed the *Ranger* system flight acceptance tests (these tests are discussed in some detail in Section III of this Report) with the exception of the heat sterilization test. The sterilization part of the flight acceptance test is separated from the others since it is accomplished fairly late in the pre-flight assembly and checkout of the spacecraft (as close to fuel-

Table 2. Nominal system pressures

Item	Nominal pressure, psia
Nitrogen reservoir, at ignition	3,000
Nitrogen reservoir, at termination (maximum duration run)	940
Propellant tank, pre-launch pressurization	275
Propellant tank, operating	320
N_2O_4 ignition cartridge, at ignition	350
N_2O_4 ignition cartridge, at termination	210
Chamber pressure, average[a]	200

[a]Represents stagnation pressure at midpoint of thrust chamber and catalyst bed.

Table 3. Nominal system temperatures

Item	Nominal temperature, °F
Nitrogen reservoir, at ignition	70
Nitrogen reservoir, at termination (maximum duration run)	−20
Propellant tank, at ignition	70
Thrust chamber wall, during firing	1800 to 1900

Table 4. Propulsion system weight breakdown

Item	Weight, lb	
Dry unserviced weight		
Propellant tank	4.0	
Propellant bladder	0.8	
Propellant tank manifold	1.1	
High-pressure reservoir	1.6	
High-pressure reservoir manifold	0.5	
Rocket engine with catalyst	2.5	
Ignition system	0.8	
High-pressure valve	0.5	
Fuel valve	0.6	
Fuel line	0.2	
Visual gages	0.2	
Pressure regulator	1.2	
Thrust plate	3.2	
Propellant tank support structure	0.5	
Misc. mounting brackets, fasteners, etc.	1.3	
Jet vane actuator support structure	0.8	
Jet vanes (4)	0.2	
Jet vane actuators	2.1	
Pressure transducers (2) and temperature transducer (1)	0.7	
Cabling	2.1	
Total dry unserviced weight		24.9
Wet weight		
Propellant (N_2H_4)	13.5	
Oxidizer (N_2O_4)	0.04	
Nitrogen gas	0.7	
Total wet weight		14.24
Total weight		39.14

ing and pressurizing as feasible, so that a minimum of time exists after sterilization). Sterilization consists of heating the completely assembled dry propulsion system for 24 hr at 125°C (257°F). After cooling, the unit is leak-checked utilizing biological filters on the various pressure ports so as not to contaminate the system. At the appropriate time in the countdown the spacecraft is moved to an Explosive Safe Area for installation of the midcourse and retro propulsion systems and pyrotechnics. As a portion of this sequence, the midcourse propulsion system is fueled and pressurized. The schedule is such that the unit is kept under surveillance for 8 to 10 hr after pressurization to verify its leak tightness. The unit is then installed in the spacecraft, the shroud is placed over the complete spacecraft, and the entire assembly is filled with a sterilizing gas atmosphere. From this time until the spacecraft is mated to the launch vehicle and the telemetry system is activated, the propulsion unit is not monitored. It is significant to note that in the typical countdown procedure this period of time may be 10 days to 2 weeks. Thus the propulsion system must stand ready, fueled, completely pressurized, and armed with explosive actuators for a significant length of time.

The in-flight operational sequence of the midcourse propulsion system is controlled by the on-board Central Computer and Sequencer (CC&S) which receives the time, direction, and magnitude for the midcourse rocket firing through the ground communication link. After the spacecraft has assumed the correct firing attitude (maneuver being executed at approximately 16 hr after launch), ignition of the midcourse propulsion system is accomplished at the prescribed time through an electrical signal from the CC&S. Inasmuch as the propellant tank is prepressurized to approximately 25 psi below the normal tank operating pressure, the rocket engine ignition can occur concomitantly with the release of the high-pressure gas to the pressure regulator without allowing time for the propellant tank pressures to build up to the normal operating level. Thrust termination is controlled by the CC&S via an electrical signal once the specified velocity increment has been realized as computed by the spacecraft integrating accelerometer. During the rocket engine firing, spacecraft attitude is maintained by the autopilot-controlled jet-vane system.

The detailed sequence of events involved in firing of the propulsion system is as follows:

1. At the command signal from the CC&S to ignite the rocket, the normally closed, explosively actuated valves 15, 8, and 4 (refer to Fig. 6 for location of these components) are activated, allowing regulated gas pressurization of the propellant tank, propellant flow to the rocket engine, and injection of a small quantity of nitrogen tetroxide to the rocket engine.

2. Hypergolic ignition ensues, followed by continuous catalytic monopropellant decomposition of the anhydrous hydrazine.

3. At the command signal from the CC&S to terminate rocket thrust, the normally open, explosively actuated valves 7 and 21 are activated, terminating propellant flow to the rocket engine and positively isolating the remaining pressure in the pressure reservoir from the propellant tank.

III. COMPONENT DESIGN AND DEVELOPMENT

A. Development of the Rocket Engine Decomposition Chamber

The heart of the propulsion system is the monopropellant rocket engine. The design of an efficient, minimum size decomposition chamber for monopropellant hydrazine requires determination of the optimum combination of injector design, catalyst bed sizing, and ignition system. During the early part of the project, approximately 125 engine firings were made, in which the following parameters were varied: (1) injector configurations, including evaluation of single-spray jet designs, multi-spray jet designs, spray coarseness, distance of injector from catalyst bed, and positions of jets in relation to reactor walls; (2) catalyst bed volumes; (3) catalyst particle size; (4) chamber pressure; (5) quantity of oxidizer used for ignition; and (6) ignition system pressure. The result of this investigation is the flight engine configuration shown in Fig. 7 and 8.

Figure 7 is a sectioned sketch of the motor showing the internal configuration of the rocket engine. Fuel is supplied through a common manifold to four atomizing spray jets. The oxidizer required for ignition is injected through a centrally located atomizing spray jet. The catalyst bed is nominally 2.6 in. in diameter and 3.5 in. in length. The catalyst used is JPL Type H-7. This catalyst is prepared from an aluminum oxide catalyst support impregnated with an equimolal solution of iron, nickel, and cobalt nitrates. The catalyst is then heated to reduce the nitrates, leaving only the base metals on the support. The catalyst particles are spherical and 3/16 in. in diameter.

Figure 8 is an external view of a partially assembled rocket engine and indicates principally the injector head, showing the integral design of the fuel manifold and oxidizer inlet tube.

The design of the catalytic reaction chamber used in this system was based upon the techniques developed by A. F. Grant, Jr. (Ref. 7). The reader is directed to this reference for general information, procedures, and design equations describing the design of catalytic reactors for use with monopropellant hydrazine. Physically the engine is of all-welded construction fabricated from stainless steel for the injector head, and Haynes Alloy No. 25 for the combustion chamber and exhaust nozzle.

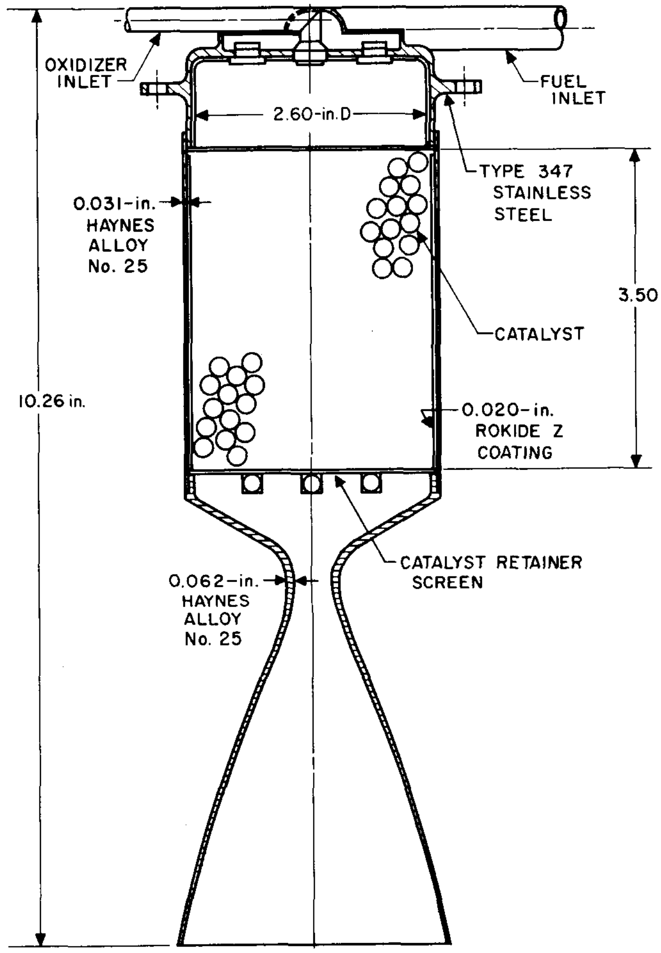

Fig. 7. Fifty-pound-thrust monopropellant-hydrazine rocket engine

B. Development of the Ignition System

The requirement of zero-gravity ignition for the propulsion system necessitated the development of a unique ignition system. Previous to the present project, considerable work was accomplished at this Laboratory toward obtaining a simple, reliable, rapidly operating hypergolic ignition system for monopropellant-hydrazine gas generators (Ref. 8). Most of the work reported described a simple system employable in positive gravity environments. The general principle of the device was to place in a small common reservoir a quantity of hypergolic oxidizer (nitrogen tetroxide, in general) and gaseous nitrogen under pressure. Outlet from the "cartridge" was sealed with a normally closed, explosively actuated valve. Energizing the valve thus allowed the pressurized mixture to be blown into the decomposition chamber. Several hundred tests of this simple system were made with consistent success. In developing the zero-gravity cartridge, the experience gained in these earlier studies was used extensively. In this case, however, a means of separating the liquid oxidizer and the pressurizing gas was required. Several variations of the system were tried, a prototype of the final version being shown in Fig. 9.

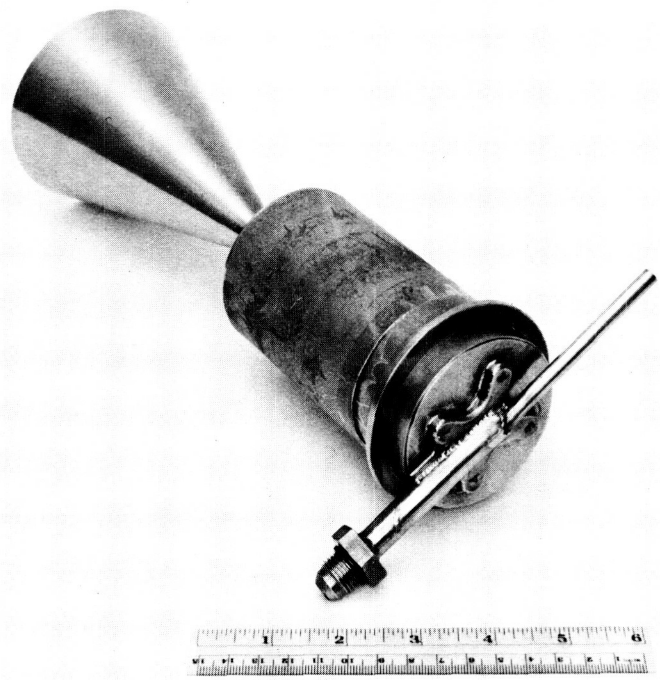

Fig. 8. Partially assembled rocket engine showing injector manifolding

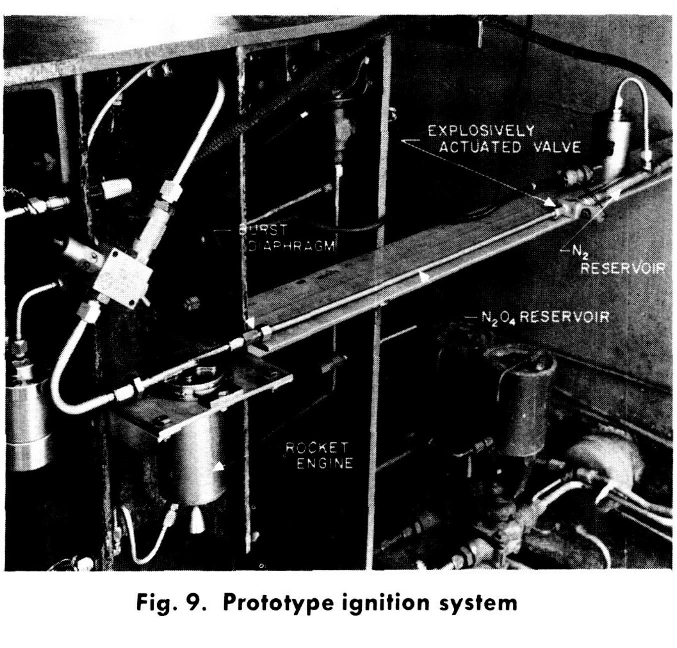

Fig. 9. Prototype ignition system

This figure shows a prototype flight engine in a typical test setup. The zero-gravity ignition system is shown at the right of the picture. From right to left, the system consists of a prepressurized nitrogen gas reservoir, a normally closed, explosively actuated valve (which is the means for separating the liquid oxidizer from the pressurizing gas), an oxidizer reservoir, and a burst-diaphragm holder. The optimum quantity of oxidizer for reliable ignition was found by experiment to be 12 cc; the optimum length-to-diameter (L/D) ratio for the oxidizer reservoir was found to be approximately 90. Upon actuation of the valve, the nitrogen gas forces the oxidizer into the chamber, and with the long L/D ratio of the tube, the gas will not by-pass around the liquid with the tube in any attitude. In tests wherein the tube was placed at various attitudes (including horizontal, the most likely attitude for failure), virtually all oxidizer was evacuated from the tube, and successful and consistent ignitions were obtained.

In the actual flight configuration, the long oxidizer reservoir was bent to fit the space available. This fact in itself is an interesting aspect of this system, i.e., the system configuration is flexible and can be adapted to varying space allocations.

C. Design of the Expansion Nozzle

The contour of the divergent portion of the rocket exhaust nozzle was optimized to deliver the maximum thrust coefficient consistent with the physical boundary conditions imposed by the spacecraft configuration and the design of the attitude-control jet vanes. An available "method of characteristics" computer program was utilized to calculate the nozzle dimensions; Mach number distribution along the wall for six divergent contour nozzles was obtained utilizing the method described in Ref. 9 for axisymmetric hypersonic wind tunnels. The thrust coefficients were computed from equations in Ref. 10. The specific heat ratio, γ, was estimated to be 1.27 for the hydrazine decomposition products. The other computation parameters were the exit Mach number, M_E, and the expansion angle, Θ_A.[1] The magnitudes of these parameters for the six cases were: M_E = 5, 6, and 7 for 90% maximum Θ_A, $M_E = 7$ for 90% of maximum Θ_A and 5% partial cancellation (a device to reduce contour discontinuity, see Ref. 9), and M_E = 8 and 9 for 90% of maximum Θ_A. The computations resulted in dimensionless wall-coordinate parameters with corresponding thrust coefficients, Mach numbers, and pressure ratios.

[1]Maximum Θ_A is equal to one-fourth the Prandtl-Meyer expansion angle, which is a function of γ and M_E (Ref. 9).

The space limitations imposed by the spacecraft interfaces and the midcourse propulsion system design configuration resulted in an allowable length for the divergent nozzle of 4.05 in. Figure 10 shows the variation in thrust coefficient, nozzle exit diameter, and dynamic pressure with exit Mach number for this nozzle length for the six contours as they appear in Fig. 10 from left to right, respectively, for each curve. From these comparisons, it was found that a nozzle diameter of 2.904 in., dynamic pressure of 5.11 psia, and Mach number 4.49, associated with the maximum thrust coefficient 1.793, would most satisfy the jet vane requirements. Thus, the nozzle contour providing these values was considered optimum for both the propulsion system performance and the jet vane design.

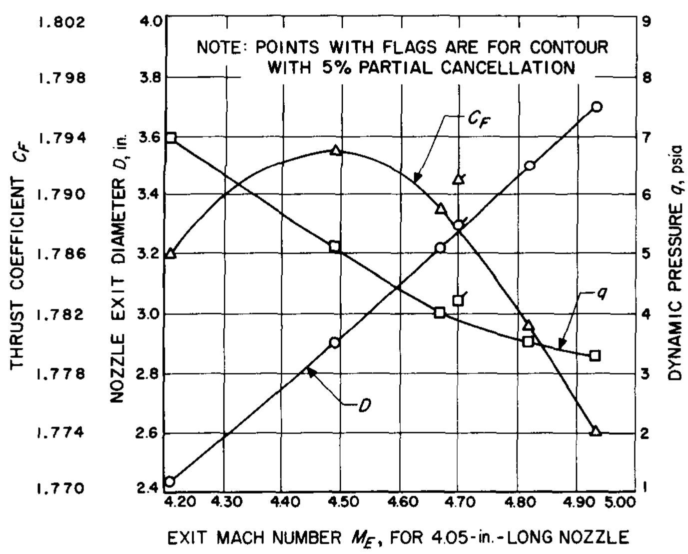

Fig. 10. Thrust coefficient, nozzle exit diameter, and dynamic pressure as a function of exit Mach number

D. Rocket Engine Testing

Subsequent to the engine firings conducted during the determination of the optimum configuration for the rocket engine catalyst bed, static test firings were made to proof-test the flight design rocket engine injector and its associated plumbing, including the complete flight design ignition system. The tests were generally satisfactory, and the engines performed as anticipated. During the tests a problem was encountered in the fuel feed line at engine shutoff. The explosively actuated valve functioned so rapidly that an instantaneous over-pressure (hydraulic hammer) resulted. This over-pressure, upon one occasion, ruptured the fuel inlet line. This problem was resolved through the substitution of a flexible, teflon-lined, wire-jacketed fuel line in place of the thin-walled aluminum tubing originally specified. This redesign allowed the line to absorb the energy of the over-pressure without rupture.

As a portion of the flight acceptance test procedure, each flight injector was water-flow calibrated to ascertain its pressure drop characteristics for later use in determining the proper setting for the gas pressure regulators. After the water calibrations, each injector was test-fired in a thrust chamber. This series of tests presented a comparison of injector efficiency and reproducibility for a production lot of seven injectors. The data were readily comparable since each injector was tested in the same thrust chamber. This chamber was constructed so that each flight injector could be slipped into the top of the chamber, held gas-tight with an O-ring in a water-cooled flange, and bolted down; the exhaust nozzle throat was also water cooled. The catalyst bed was changed for each test. In this way, it was felt that variations of parameters other than those associated with the injector could be minimized. The water-cooled nozzle throat made it possible to ascertain throat area accurately for each of the static firings (not possible with flight nozzles since they are uncooled and, therefore, subject to thermal variations). The average characteristic exhaust velocity obtained for the seven injectors varied by $\pm 0.75\%$.

After this series of tests the flight engines were welded and machined to their final flight configuration.

As indicated previously, the total impulse of the propulsion system is variable. The low thrust-to-weight ratio of the midcourse propulsion unit to the over-all spacecraft necessitates that a sensitive accelerometer be used. Some concern existed early in the program as to whether the random vibration and noise produced by the operation of the rocket engine and induced into the spacecraft structure might produce spurious signals to the accelerometer and cause it to count improperly. In view of this, a series of tests was undertaken in which a test model of the *Ranger* spacecraft, weighted to simulate the flight unit, was elastically suspended in an acoustically lined chamber (Fig. 11). A test rocket engine, mounted as it would be in the final flight system, was placed aboard. Flexible propellant lines were used to introduce the fuel. A flight model of the accelerometer system was mounted in its proper location in the spacecraft, and two tests of 1-min duration each were made. The accelerometer system sensitivity was such that if 1 g of noise were present, erroneous counting would occur. The actual noise output of the rocket engine as mounted was found to be 50×10^{-3} g in the first test and 55×10^{-3} g in the second test. Thus, it is evident that the vibration induced by the propulsion system would have no effect on the accuracy of integration of the accelerometer system. This low level of combustion noise is worthy of note since it is a very desirable characteristic of this simple monopropellant system.

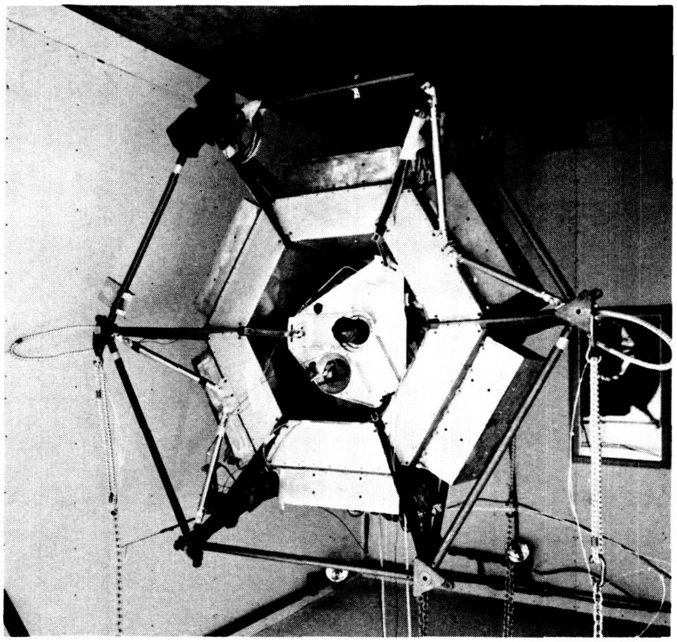

Fig. 11. Rocket test setup in acoustical chamber

E. Development of the Propulsion System Tankage

During the *Ranger* preliminary design program the intent was to utilize propulsion system tanks as large as was consistent with the volume available for the propulsion system in the spacecraft. It was hoped that excess propellant, above that actually required to attain the desired velocity increment of the mission, could be added at the time of final spacecraft assembly as added insurance to cover cases of larger injection errors. As the design of the spacecraft proceeded, it appeared that this desirable feature could not be accommodated from a spacecraft weight standpoint and, therefore, carrying oversized tankage was not practical. The velocity increment for the *Ranger* midcourse maneuver was specified in the design criteria to be 120 ft/sec. These data and consideration of propellant reserves and specific impulse loss due to the jet vanes in the exhaust nozzle determined that, for an expected spacecraft weight of 750 lb, the propellant weight should be 13.5 lb. It was further concluded that both the high-pressure reservoir and the propellant tank would be fabricated of titanium alloy (6% aluminum, 4% vanadium).

The basic design philosophy of the spacecraft required that personnel be in the close vicinity of the propulsion system tanks when the tanks were pressurized to full operating pressure. The design of the vessels was based upon Laboratory specifications regarding minimum safety factors to be used in vessels pressurized in the vicinity of personnel. In all cases, these specifications are more stringent than the USAF safety regulations in effect at the Atlantic Missile Range. The minimum safety factor used in the design of the tanks was that the burst pressure be 2.2 times the maximum operating pressure. The pressure reservoir tank is spherical in shape, and the propellant tank is an oblate spheroid (Fig. 5). The tank shapes were dictated by space limitations in the spacecraft.

As indicated in Section II of this Report, the propellant tank is prepressurized to reduce the duration of starting transients. This feature, coupled with the requirement for the fueled and pressurized system to be handled safely at a maximum pre-launch temperature of 165°F, necessitated that a rather large ullage be designed into the propellant tank. The combined temperature effects of the volumetric expansion of the fuel, as well as the prepressurization gas pressure increase, raised the tank pressure at 165°F to excessive pressure levels for conventional ullage volumes. A design analysis indicated that an ullage volume of 15% was desirable. With this value a maximum tank pressure of 460 psia was computed.

Initial models of the completed tanks were subjected to a proof test program consisting of several combinations of vibration and pressure cycling. Several units of each different tank design were ultimately taken to the burst point. In every case, the burst pressure was above the minimum design safety factor. Test units of the spherical high-pressure reservoir, which was designed for a 3,600-psi maximum operating pressure, burst at 10,200 to 10,300 psi, a safety factor of 2.85. For the propellant tank, with a maximum operating pressure of 460 psi, burst pressure was 1,840 to 1,854 psi, a safety factor of 4.0. The high safety factor results from conservative use of the less well-established design criteria for the oblate spheroid shape as compared with those for the spherical shape. All flight tanks were subjected to a flight acceptance test consisting of three cycles of hydrostatic pressure to approximately 75% of burst pressure and one cycle of high pressure at liquid nitrogen temperature.

Shortly after initiation of the fabrication of the tankage, reduction in the allowable total weight of the spacecraft reduced the midcourse unit propellant allocation to 12.3 lb. Subsequently, however, two system changes appeared to permit an increase to 14.6 lb in the amount of propellant carried in the midcourse propulsion unit. The changes were (1) an increase in the allowable total weight of the spacecraft owing to changes in the booster system and (2) a reduction in the design specification for maximum pre-launch temperature from 165 to 125°F. Calculations indicated that the increased quantity of propellant could be carried with no change in tank size and no decrease in the tank safety factor since the decrease in maximum pre-launch temperature allowed a decrease in required ullage space in the tank and, therefore, maintenance of the same maximum pressure.

F. Development of the Propellant Expulsion Bladder

One of the most interesting design and development aspects encountered in the midcourse propulsion system was in the fabrication of the bladder for the propellant tank. This device consists of a flexible butyl rubber bag which contains the propellant and maintains a positive separation from the pressurizing gas (Fig. 12). A unique feature of the design, resulting from space and weight limitations in the spacecraft, involved the requirement that the pressurizing gas inlet and the liquid hydrazine outlet be at the same end of the tank. This necessitated that the gas inlet be accomplished through an annular ring around the fuel outlet and that the bladder be constructed with ribs on its external surface so that a free passage was always available for the gas to pressurize the tank uniformly. For compatibility and reliability reasons, the bladder could not contain any seams, but was required to be of one-piece construction. The bladder was fabricated of a butyl rubber which has proven in the past to be compatible with anhydrous hydrazine. The butyl compound used in this application was Fargo Rubber compound No. 6-60-26.[2] The material is considered to be satisfactory for several months at design conditions encountered in this device. The design restraints proved to be an extremely difficult problem for rubber fabrication, and necessitated the development of unique tooling and many sample fabrications before a satisfactory unit could be made by the subcontractor.

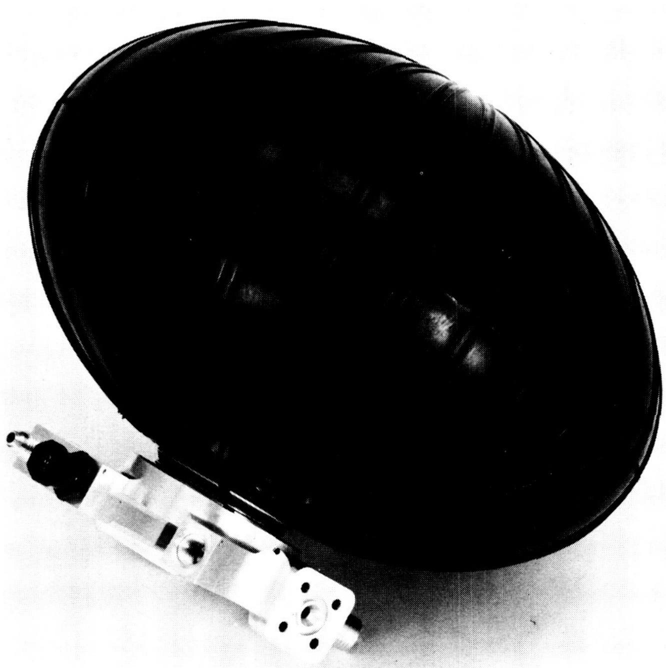

Fig. 12. Propellant tank bladder attached to the tank outlet fitting

G. Development of the Pressure Regulator and Valves

Detailed design of the gas pressure regulator and the explosively actuated valves was subcontracted to commercial vendors.

The gas pressure regulator was manufactured by Sterer Engineering and Manufacturing Company[3] according to Laboratory specifications and packaging requirements. The regulator design parameters are shown in Table 5, and an outline drawing of the unit is shown in Fig. 13. As is evident from Fig. 13, the outlet side of the regulator contains a bellows; this packaging of the regulator was unusual in that the unit itself was employed as the plumbing link between the high-pressure reservoir module and the propellant tank module. The bellows was employed to separate the "tie down" of the modules so that the system would survive the vibrational environment encountered during booster firing. The relative position of the regulator in the system is shown in Fig. 14.

The remaining valves utilized in the system are all of the explosively actuated type. All of the units were manufactured by the Conax Corporation.[4] Physically, three different designs were employed. In the fuel circuit a normally open and a normally closed valve were combined into a single body, the upstream part of which was bolted directly to the propellant tank outlet fitting. The downstream part connected directly to the engine flex line inlet. The valve and its location are shown in Fig. 15.

[2]Fargo Rubber Corporation, 137 East 58th Street, Los Angeles 11, Calif.

[3]Sterer Engineering and Manufacturing Company, 11423 Van Owen Street, North Hollywood, Calif.

[4]Conax Corporation, Explosive Products Division, Buffalo, N.Y.

The high-pressure circuit utilizes a similar set of valves contained within a single body. In this case the unit bolts directly to the high-pressure tank manifold. Both inlet and outlet are side by side on the same side of the valve. This valve combination is shown in Fig. 16. The oxidizer start valve used in the ignition system is shown in Fig. 14. This valve is a normally closed unit.

Concurrent with the type-approval testing of the propulsion system tanks, the pressure regulator and the explosive valves were put through an equally rigorous type-approval test program consisting of vibration, shock, and temperature in accordance with Laboratory environmental specifications for *Ranger*. All components passed the tests satisfactorily. Of special interest is the fact that the regulator was temperature-soaked at 257°F for 24 hr as it would be in the sterilization procedures, and subsequently operated with no indication of performance degradation.

Table 5. Gas pressure regulator specifications

Inlet working pressure, psig	
Maximum	3,600
Nominal	3,000
Minimum	2,700
Outlet working pressure, psig	
Maximum	325
Nominal	300
Minimum	275
Inlet test pressure, psig	
Proof	5,400
Burst	9,000
Leak	3,200
Specified leakage, std. cc He gas/hr	
Internal	5
External	10
Rated flow	
0.002 lb/sec He gas at 300 psia	
Max. Δp at rated flow 50 psi of He gas	
Temperature range	
0 to +200°F.	
Working fluid (helium) −110 to +165°F.	
Regulator to be subjected to "slam" start, i.e., inlet pressure rise, 0 to 3,600 psia in approximately 5 msec or less; outlet pressure, 100 psia.	
Regulator to hold (helium) pressure at leakage rates specified, for periods up to 1 mo.	
Regulator to be capable of raising a 15-cu-in. ullage volume to setting level (±2.5%), from a pressure level 100 psi below setting level, in 0.25 sec max.	
Regulator to maintain setting level (±2.5%) when inlet pressure decays from 3,600 psia to 50 psi above setting level.	
Regulator to be able to repeat within the ±2.5% limits for a minimum of 6 full-duration operational cycles.	

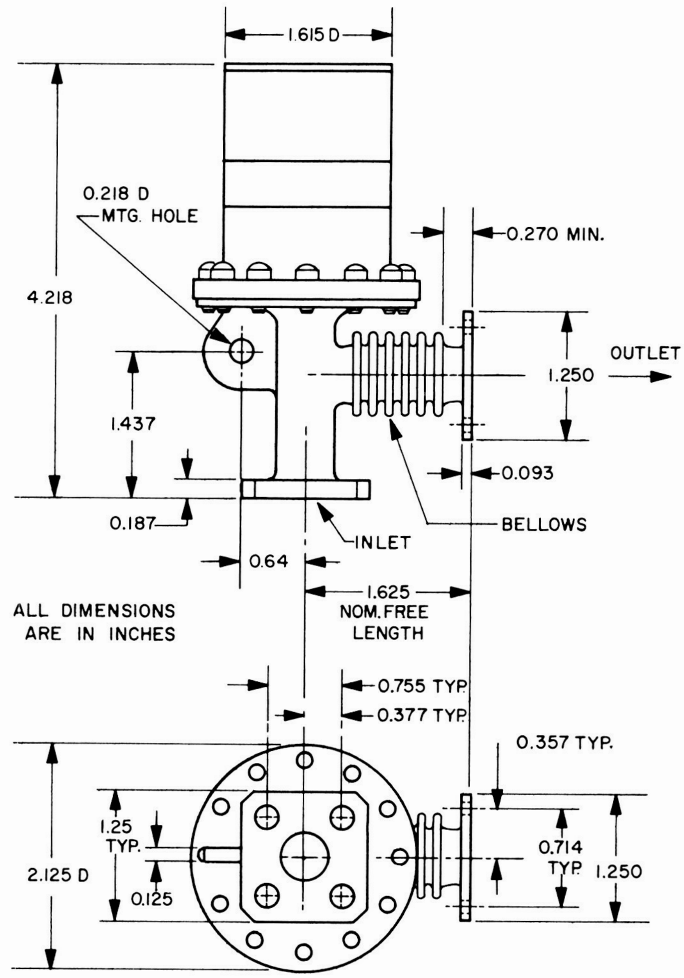

Fig. 13. Gas pressure regulator

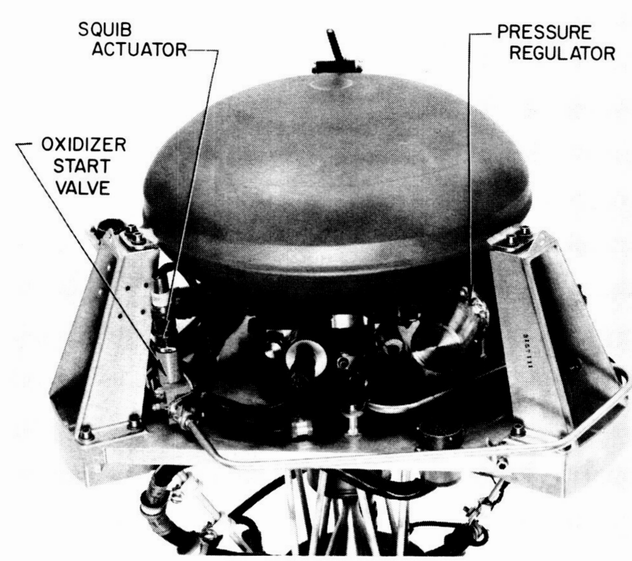

Fig. 14. Mounting position of the pressure regulator in the system

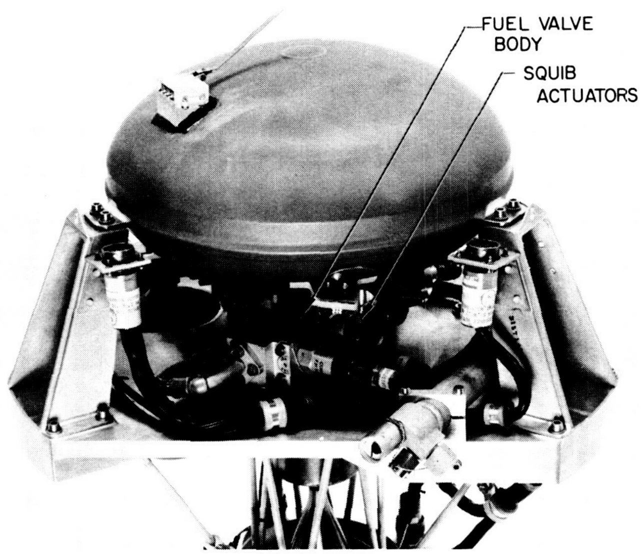

Fig. 15. Location of the combined normally open–normally closed fuel valve

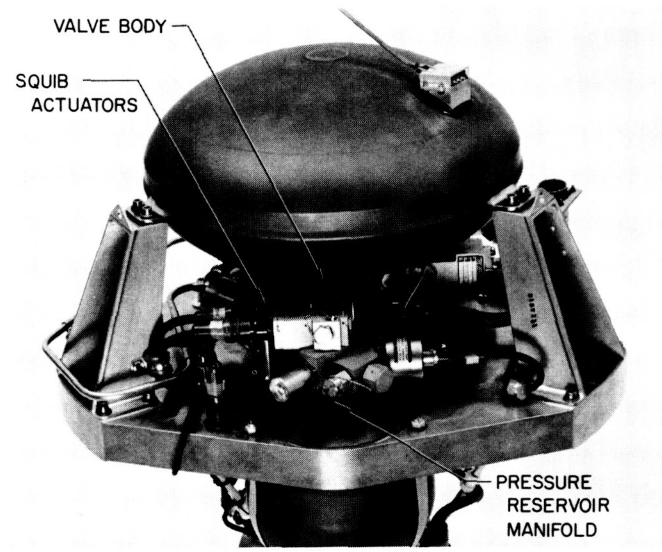

Fig. 16. Location of the combined normally open–normally closed high-pressure gas valve

IV. SYSTEM TESTING

A functional systems test model of the propulsion unit was constructed in the test cell and is shown in Fig. 9. This unit was used to study system starting transients and shutoff transients, effects of prepressurization level on such transients, verification of system pressure drops, the determination of minimum impulse capabilities of the system, and the effects on the system of complete propellant runout. The procedure in each of the tests was to simulate actual system operation as closely as possible. Explosively actuated valves were used throughout the test system; the system was fueled and prepressurized as in the actual system, and the firing procedure consisted only of one electrical signal to fire the necessary explosive valves at ignition and one signal to terminate the system at shutoff.

In the tests concerned with start-up transients at various prepressurization levels in the fuel tank and ignition cartridge, pressure levels chosen were equivalent to those which would exist in the pressure reservoir, fuel tank, and ignition system at three different temperatures: the design minimum operating temperature, 40°F; nominal operating temperature, 70°F; and maximum operating temperature, 165°F. In all cases, successful ignition and satisfactory operation were attained. In addition, one test was made to determine the effects of propellant runout. The system was started normally and the engine valving was left open until the propellant was completely exhausted, thus stopping the motor. No problems were encountered with this procedure, and it would appear that no explosions or damage would occur in flight if the shutoff signal were not given by the spacecraft. However, if the signal were not given, the wrong velocity increment would be delivered to the spacecraft.

With the receipt of complete sets of flight hardware, a series of system tests culminating in static firings was undertaken to evaluate the entire flight system in its final packaged condition. The objectives in these tests were to determine system response rates, transients, and regulator characteristics when the regulator was close-coupled with the other system components. The initial tests of this series consisted of expelling water from the bladdered propellant tank to observe whether the technique employed was suitable to hold the bladder onto the tank manifold and to determine propellant hold-up as the bladder collapsed. In the course of these tests, it was found that the tank outlet baffle required redesign to assure that the bladder would not collapse incorrectly and trap a large quantity of fuel. The initial design consisted of a raised grating in the shape of a spherical segment. It was found that under various conditions of negative and positive gravity (accomplished by placing the tank at various attitudes) flow blockage occurred in some instances. A redesign was accomplished by placing a cantilevered pylon with a ball on top at the tank outlet, thus leaving an annular series of holes at the pylon base as a propellant drain. This design apparently forced the bladder to collapse more uniformly and assured an open channel to all parts of the bladder until it was virtually empty. The redesign was tested thoroughly and produced a propellant hold-up of less than 1%.

Next, the propellant tank circuit was mated to the helium pressurization circuit, and pressure regulator and water pumping tests were made using the various pressurization levels in the propellant tank which would be present at the environmental temperature extremes. An explosively actuated valve was used to release the high pressure gas as would be the case in the flight unit. Good results were obtained, except for the performance of the gas filter which was positioned between the gas blocking valve and the gas pressure regulator. It was found that the explosively actuated blocking valve opened so rapidly that the filter was subjected to a transient pressure differential of sufficient magnitude as to collapse the unit. The unit was redesigned after the initial test and performed satisfactorily thereafter. After establishing the characteristics and acceptability of the pressurization circuit, this system was mated to the rocket engine and ignition systems. The system was fueled and pressurized as it would be in the field. A series of three static firings was made. Each test was of 55-sec duration. The over-all system functioned well, all system pressures and temperatures being within tolerance. Photographs of the actual flight system are shown in Fig. 14, 15, and 16.

One of the most critical items in establishing the over-all performance and operating characteristics of the flight propulsion system is its actual operation in a vacuum environment. Because this Laboratory does not possess suitable facilities for such tests, vacuum facilities were contracted on a rental basis. The rented facilities allowed the rocket engine to be fired continuously into a steady ambient pressure equivalent to an altitude of approximately 100,000 to 130,000 ft. This pressure was sufficiently low to allow the rocket engine nozzle to flow full.

A total of 27 static firing tests was made at the off-Laboratory leased facility. The tests were divided into three general categories: (1) tests aimed at defining rocket engine performance (these tests employed a flight design rocket engine, ignition system, and fuel valving system, but utilized an externally located fuel supply system); (2) tests directed toward the calibration of the jet vanes (these tests utilized the same type of equipment as the first tests, with the addition of the jet vane system mounted below the engine); and (3) tests of the complete flight propulsion system.

On an over-all basis, the following statements described the tests:

1. Ignition was achieved without difficulty in all cases. The starting transient appeared virtually identical with that obtained under normal atmospheric pressure.

2. Engine operating characteristics (i.e., chamber pressure roughness, chamber wall temperature, and catalyst life) appeared to be identical with those in tests at atmospheric pressure.

3. The jet vanes and electromechanical jet vane actuators can withstand the high temperature environment at the rocket engine nozzle exit for the maximum run time (approximately 61 sec) without degradation.

4. The entire system operated under vacuum without mishap. No excessive heat transfer was noted to any of the close-coupled components. Operating characteristics of the pressurization system were identical with those at atmospheric pressure.

Perhaps the most severe environmental condition encountered in the spacecraft flight from a structural standpoint is the vibration environment induced during firing of the booster stages. Accordingly, a series of tests was conducted to simulate such conditions. A complete propulsion unit was mounted in a prototype *Ranger* spacecraft which, in turn, was placed in a vibration tester. Physically, the propulsion unit was fueled with water to obtain the proper flight weight, and was pressurized to approximately 100 psig (low enough to afford large safety factors for personnel in the vicinity during vibration tests). The test procedure consisted of exciting the unit and spacecraft with a sinusoidal vibration program from approximately 40 to 1500 cps at various *g* levels with vibration at the points of resonance being maintained for extended time periods. In the course of these tests, two problems were encountered. A failure occurred in the structure that supports the jet-vane actuator assembly below the rocket engine. This support, which is essentially a cantilevered structure, incurred weld cracks. Also, an abnormal excursion of a portion of the oxidizer ignition system was observed which, although it did not result in any failure, was felt to be a potential reliability problem. A modified design of the jet-vane support structure, consisting of the addition of three diagonal stabilizing struts, was introduced, and the oxidizer ignition system was supported with an additional tie-down. The tests were repeated without incident. The propulsion unit was then subjected to the *Ranger* type-approval test vibration specification in three planes. After the tests the system was visually inspected and leak checked. No indications of system degradation were found. In view of the success of these tests and the desirability of reducing the weight of the spacecraft, modifications to minimize this weight were made to the support structure that holds the propulsion unit in the spacecraft. This was accomplished and the type-approval test sequence was run again. The unit once again successfully withstood the vibration environment without leakage or damage.

One of the fundamental design criteria of the *Ranger* 3, 4, and 5 spacecraft is the requirement of sterility. At the present time the most desirable method of assuring sterile equipment is high temperature. The approved Laboratory standard is 257°F for 24 hr. In addition, ethylene oxide gas diluted with freon is pumped into the spacecraft shroud prior to launch. Thus, the flight equipment must withstand high temperature for long duration internally and externally, as well as withstand the effects of the gas mixture externally.

A complete unit empty of liquids or gases was placed in an oven and soaked for 24 hr at 257°F. The unit was cooled and then placed in a chamber filled with the ethylene oxide sterilizing gas mixture for another 24 hr. After these tests the unit was visually inspected, leak tested, fueled, pressurized, and statically fired. No abnormalities were noted in the physical appearance of the unit or in its operation.

V. FLIGHT OPERATIONS

A. Ranger 3

Ranger 3 was launched on January 26, 1962. Although the flight of *Ranger 3* did not fulfill its complete mission objectives of lunar impact, it is apparent from telemetry data that the midcourse propulsion system performed satisfactorily during the commanded midcourse maneuver sequence.

Pre-launch preparation of the midcourse propulsion system at the Atlantic Missile Range consisted of two phases. The propulsion system was shipped across country installed in the spacecraft. After a cursory inspection of the entire spacecraft, the propulsion unit was dismounted from the spacecraft system and subjected to a complete visual inspection, and the tankage and plumbing were checked for leaks. The unit was then placed in a sterilization oven and the entire assembly was subjected to 257°F for a 24-hr period. After the unit was removed and cooled, it was again leak-checked. At this time a minute leak was found in the propellant tank bladder. This difficulty necessitated that the tank be torn down and a spare bladder substituted. After verification of the repair, the tank and bladder assembly were resterilized.

No additional problems were encountered and the unit was returned to the spacecraft for a final system test sequence. The system was partially pressurized so that the telemetry gages could be checked out during the dummy runs. Several days later the unit was returned to the propulsion facility for final preparation, fueling, and pressurization. Loading and pressurization of the ignition system proceeded without incident. However, when the fuel tank was loaded with anhydrous hydrazine, a problem of fuel leakage across the propellant tank bladder was noted. Inspection indicated no leaks in the bladder itself but pointed to the probability of a leak occurring at the point where the bladder is attached to the tank outlet fitting. It became apparent that the difficulty stemmed from the change made late in the development program which increased the amount of fuel loaded in the tank from the initial design value of 13.5 lb to 14.6 lb. This change (which was desirable but not mandatory) was made to take advantage of an increased weight allowance available in the spacecraft in order to furnish an excess in maneuver capability beyond the design criteria. Preliminary tests made with prototype hardware had indicated that the change could be accommodated without difficulty. However, upon close inspection in the field it was concluded that the bladder wall thickness in the production item was very close to the upper tolerance limit, a condition not present in the prototype; as a result, sufficient stretching of the bladder was not possible without deforming and reducing the bladder neck wall thickness where the bladder was attached to the tank outlet fitting. It was found necessary to return to the quantity of fuel originally specified. Actually this situation still produced a velocity increment greater than the original design requirement, inasmuch as the overall spacecraft weight was less than that of the initial design. In the final operation, the system as loaded had a velocity increment capability 12.5% greater than that of the original design (120 ft/sec) rather than an anticipated 20% greater capability. The remainder of the loading and pressurizing sequence proceeded without difficulty. One additional change was made wherein nitrogen gas was substituted for helium gas in the reservoir to take advantage of the excess weight available by increasing reliability with regard to leakage of gas. Preliminary tests had indicated that such a substitution was not detrimental to the regulator.

After verifying that the unit was pressure tight, the system was installed in the spacecraft. It is interesting to note at this point that the fueled and pressurized unit remained in the spacecraft for 12 days prior to launch. During the countdown the readiness of the system was verified by monitoring the pressures and temperatures of the gas reservoir and the fuel tank. In-flight telemetry data indicated no degradation in pressure levels or change in temperatures occurring in the system for approximately the first 4 hr. After this time it was noted that the entire spacecraft was gradually rising in temperature. The telemetry data recorded during flight are shown in Fig. 17. This increased temperature condition was reflected in the gas reservoir pressure, which rose somewhat. Fuel tank pressure did not show a comparable increase, but this was felt to be due to the more central and isolated position of the fuel tank in the spacecraft.

The midcourse maneuver sequence was undertaken approximately 13½ hr after launch, with the actual rocket firing occurring 27 min after the start of the maneuver positioning. From post-firing data it appeared that the rocket fired and shut down in a normal manner. Unfortunately, no telemetry data were obtained during the midcourse sequence because of radio transmission problems; hence the details of the actual rocket operation must be deduced from data taken before and after the maneuver. After firing, the pressures and temperatures appeared to be as expected, with the gas reservoir temperature showing the effects of thermal radiation from the rocket engine. Although data obtained for the next several days were erratic at times, it appeared that the tank pressures remained constant, indicating the continued leak tightness of the system.

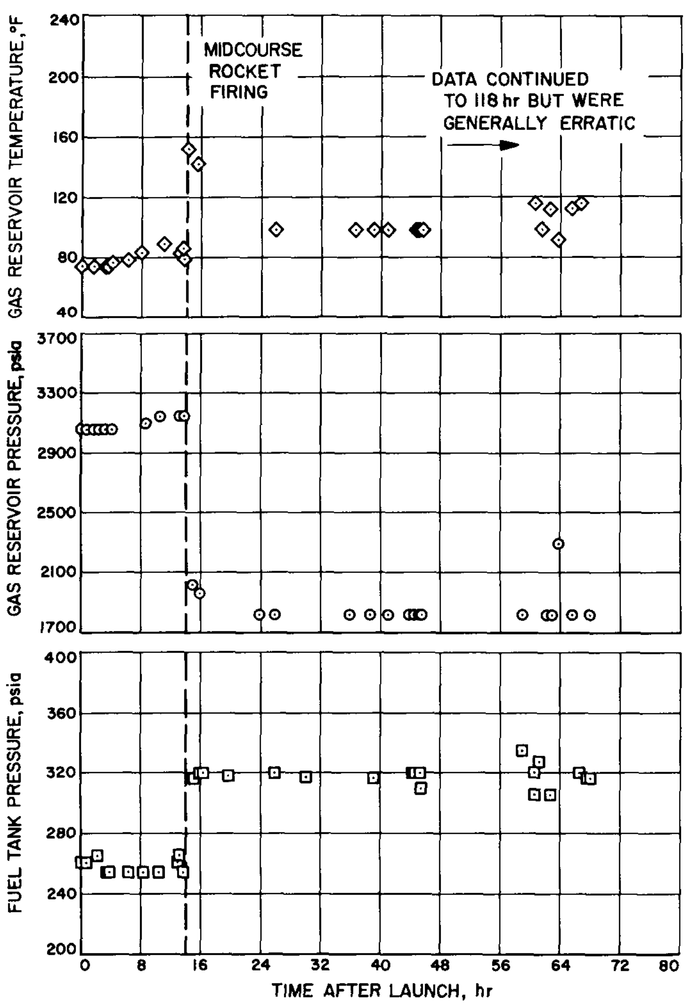

Fig. 17. Telemetered midcourse propulsion system data from *Ranger 3*

Consistent with the loss of propulsion system operational information during the midcourse maneuver sequence, the exact velocity increment produced by the propulsion system could not be determined from the telemetry records. However, it was noted that the duration of the tracking station recordings of the amount of the doppler shift during the firing of the rocket agreed very closely, within a few tenths of a second, with the anticipated rocket burning time based upon nominal performance values obtained in the development program. Hence it would appear that the actual specific impulse of the unit was very close to the design value.

B. Ranger 4

Ranger 4 was launched on April 23, 1962. Within a few minutes after lift-off a failure occurred on board the spacecraft which rendered it incapable of orienting itself in space, receiving or acting on commands, or telemetering commutated data back to Earth. The injection trajectory upon which the spacecraft was placed by the booster stages, however, was such that even without the use of a midcourse maneuver the craft impacted the moon.

The midcourse propulsion system pre-launch operations were accomplished without difficulty. For this launch the propulsion system checkout sequence was altered somewhat in that the unit was heat-sterilized and leak-tested at JPL rather than at the Atlantic Missile Range. Upon receipt of the spacecraft at Cape Canaveral and after the usual inspection of the entire system, the propulsion system was removed and again leak checked. After its acceptability was verified, the unit was partially pressurized so that the telemetry gages could be checked during the subsequent dummy runs. After several days the unit was again removed from the spacecraft and at the proper time in the pre-launch operations it was fueled and pressurized without incident. The loaded unit was placed aboard the spacecraft and the system was gas-sterilized. At the requisite time in the countdown the spacecraft was placed aboard the booster rockets. After the telemetry link was established a problem was found to exist in the high-pressure gas reservoir pressure transducer. The problem consisted of an intermittent electrical contact. After a thorough evaluation it was decided to accept this condition with the expectation that at least partial data would be available. No further problems were encountered with the propulsion unit through launch. A small amount of telemetry data was obtained after lift-off and prior to spacecraft failure. These data indicated that the high-pressure reservoir and fuel tank were holding pressure at the proper level. Since the failure prevented the spacecraft from accepting commands, no midcourse maneuver was performed.

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